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ELECTRIC PROPULSION FOR AN INTERPLANETARY ASTROPHYSICS MISSION

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Abstract

The Energetic Transient Array (ETA) is a mission, proposed by the Center for Space Research (CSR) at MIT, to set up an interplanetary constellation of six microsatellites for Gamma Ray Burst (GRB) astrometry. The microsatellites will be deployed into distinct heliocentric orbits by a carrier spacecraft which will be propelled by a stationary plasma (SPT-70) electric thruster. This paper presents a discussion of the issues related to the application of electric propulsion technology in a primary propulsion mode. An overview of the mission is given. The results of a trade study, performed to assess various propulsion options, show that the SPT-70 is best suited to meet ETA system requirements. A constellation deployment scenario has been developed around the operation of two SPT-70 thrusters in serial mode. Details of SPT-70 system hardware are presented together with system block diagram and configuration. ETA is a unique opportunity to demonstrate the operation of electric propulsion technology in an interplanetary setting and in the mode of primary propulsion.

Overview

The Energetic Transient Array (ETA) is a low cost, multiple microsatellite mission, proposed by the Center for Space Research (CSR) at MIT, to set up a dedicated interplanetary network for Gamma Ray Burst (GRB) astrometry [1]. Six microsatellites, each equipped with four GRB detectors, will be deployed into distinct ~1 AU heliocentric orbits by a carrier spacecraft which will be propelled by a stationary plasma electric thruster (SPT-70).

Gamma Ray Bursts (GRB's) are flashes of soft gamma radiation with durations ranging from milliseconds to minutes, which arrive at the Earth about once a day with a roughly isotropic distribution. If originating from outside our galaxy, their sources must be among the most powerful events in the universe. The distance, origin and nature of GRB's are issues which have remained unresolved ever since the first GRB measurements were made in the late 1960's, making them one of the most enduring and perplexing scientific mysteries of the latter half of the 20th century. Accurate GRB source localization and timely dissemination of data are necessary to test models and theories of GRB sources [2,3]. The positions of GRB's are determined by measuring their times of arrival at distinct spacecraft separated by interplanetary distances. Over a period of 2 years, the dedicated ETA will localize about 800 GRB's to higher accuracies than previously achieved by any other observations to date. Accuracies of up to 1 arc second will be achieved, compared to 1 arcminute from the best current GRB astrometry systems.

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Figure 1 illustrates how the ETA system will function. When a GRB occurs (1), the detectors on the microsatellites will be triggered and GRB time profiles will be stored in the onboard solid state memory. Two "trigger" satellites, in orbits within close proximity of Earth (0.02-0.04 AU), will have sufficiently high datarates to downlink the full burst profile in real-time (2). The purpose of this is to provide a near real-time alert capability to initiate ground-based operations. Ground processing will determine which of the burst profiles downlinked by the trigger satellites is interesting and a GRB template will be prepared (4) for uplinking at high data rates to the other microsatellites (5,6). Template correlation will be carried out onboard to determine the times of arrival (7) and only this data and a correlation confidence factor need then be telemetered back to Earth for localizing the GRB (8,9). Derived GRB coarse coordinates (10) can be relayed to other observatories such as the Hubble Space Telescope for follow-on observations (11,12).

The constellation of 6 microsatellites needs to be deployed in an optimal manner to achieve the minimum distribution necessary for science activities to commence in the time specified by program requirements. In order to achieve this, the microsatellites will require large velocity differences (ΔV) relative to each other to achieve the constellation "spread". This, in unison with the cost limit (and hence mass limit) imposed on the program, requires a propulsion system with a high specific impulse (Isp) for the deployment spacecraft (carrier spacecraft), in order to maximize ΔV while minimizing launch mass. Electric propulsion (EP), with its inherently high Isp, is an ideal candidate for the propulsion system of the carrier spacecraft. If approved, ETA is scheduled for launch in the year 2000 aboard the DeltaLite launch vehicle.

This paper presents an overview of the issues related to the application of EP technology to the ETA mission. A top level trade study performed to select the propulsion system is described. Details of the operational strategy for the EP system are given in the summary of the ETA baseline mission scenario. The configuration of the propulsion system and details of hardware elements are included with a discussion on application issues associated with the SPT-70 system.

Propulsion Options/Trade Study

This section summarizes the top level trade study that was perform to assess the suitability of a number of propulsion alternatives for meeting system propulsive (ΔV) requirements [4].

The most important system requirements were those associated with constellation "spreading" rate, number of microsatellites within the constellation and launch capabilities of the DeltaLite launch vehicle. While it is ideal to achieve long inter-spacecraft baselines in the shortest time, this imposes high ΔV requirements on the propulsion system. A constellation angular "spread" requirement of ~90 °/yr (at ~1 AU from the Sun) was defined as a compromise between scientific operational capability and propulsion ΔV requirements. This requirement also took into account programmatic funding availability, with the aim of maximizing scientific return for the dollar. Scientific operations require a minimum of 4 microsatellites but reliability/redundancy considerations necessitate more spacecraft. Constellations of 4-8 microsatellites were considered, the upper limit imposed by GRB detector, which dictate the microsatellite minimum mass limits. The DeltaLite launcher can inject ~570 kg to escape (C3=0) [1] and this was used as a baseline. In order for this scientific mission to be selected, it is imperative that technical risk be minimized.

To this effect, it is important that the selected propulsion system be highly reliable, have substantial flight experience and be readily available.

A set of criteria was identified to assess the propulsion options. The major ones were propulsion system performance, flight experience, availability, cost, reliability/redundancy and operational flexibility. Performance was quantified by the total system launch mass required to meet the system angular "spread" requirement. This required translating the "spread" requirement into an equivalent ΔV requirement for the propulsion system. Through consideration of Keplerian orbit equations, the relationship between constellation angular spread and required ΔV is given by [4]

$$\Delta V_{NET} \approx \frac{\overline{a}\,\Delta\vartheta}{3\,T_s} \qquad (1)$$

where ΔV net is the total ΔV requirement, $\overline{a}$ is the "average" value of the semi major axis of the orbit, $\Delta\vartheta$ is the required angular spread and T_s is the time to attain the required angular spread. For achieving a spread rate of 90°/yr, and assuming orbits of ~1 AU, the net ΔV required is approximately 2400 m/s. This is the differential velocity, relative to first microsatellite, that must be imparted to the last microsatellite in order that the angular separation between them be about 180° after 2 years plus thrusting time. From the perspective of a scientific mission such as ETA, the propulsion system has also got to be highly reliable, well proven in space and be readily available; these criteria were deemed to be equally important with propulsion system performance.

Microsatellite mass was conservatively estimated to be 50 kg and the trade study considered two different schemes for deploying the microsatellites. These were microsatellites with their own propulsion systems (dedicated microsatellites) or microsatellites deployed by a carrier spacecraft with a propulsion system. It was assumed that an "adaptor" spacecraft with requisite stacking/deployment, guidance, telemetry systems and would be required with a mass of about 70 kg. This "core" mass would be supplemented by propulsion system mass and associated solar array and power processing mass required for a carrier spacecraft scheme.

The advantages of dedicated microsatellites include redundancy and protection against single point failures that could potentially plague a single propulsion stage on a carrier spacecraft. This has to be traded off against available launch mass, system cost and microsatellite design complexity. A carrier spacecraft scheme allows simpler microspacecraft design in this regard, with the additional advantages of allowing some flexibility in mission design as well as providing a large platform in interplanetary space for post-deployment operations. The obvious disadvantage of the carrier spacecraft is that a larger mass has to be accelerated to the final ΔV.

Propulsion options for the dedicated microsatellites included Solid Rocket Motors (Thiokol STAR 13 SRM), monopropellant Hydrazine, bipropellant (Hydrazine/N_2O_4) and a 100 W Hall plasma electric thruster. The 100 W Hall thruster is currently unavailable and was included for comparison purposes only. Thrusting was assumed to be aligned with the heliocentric velocity vector, except for the SRM option where a constant propellant load was assumed and the ΔV distribution was attained by angling the thrusting with respect to the heliocentric velocity vector. For the carrier spacecraft scheme, a chemical bipropellant stage (Kaiser Marquardt R4D), the OAC MR 508 arcjet, the ISTI SPT-70 plasma engine

and the Hughes Xenon Ion Propulsion (XIPS) ion engine were assessed. The mass of the dry propulsion systems were estimated as follows: For the monopropellant system, from Ref. [12],

$$M_{P.S.}\ (kg) \cong 18.9+0.05\ Mp\ (kg)\ .$$

For bipropellants, also from Ref. [12],

$$M_{P.S.}\ (kg) \approx 23.2+0.04\ Mp\ (kg)$$

For Hall and ion engines, we took advantage of our own preliminary design work on the SPT-70 system to develop the expression

$$M_{P.S.}\ (kg) \cong 72\ P(kW) + 0.335\ Mp\ (kg)$$

which includes the solar array, power conversion equipment, flow system and engine (plus gimbals). For the arcjet, we used a hybrid expression which combines the mass/kw of other EP thrusters and the 0.05 kg (kg fuel) of the monopropellant system. Finally, the solid rockets were assumed to require 4.65 kg casing mass (STAR 13), plus an equal mass for attachments. The ΔV distribution for the carrier spacecraft scheme was defined such that the first microsatellite was deployed, after Earth escape, without any ΔV, and subsequent deployments were spaced to ensure equal differential ΔV's so that the microsatellites are uniformly distributed in azimuth. For the dedicated microsatellite propulsion options, one microsatellite is given a small "insertion" ΔV of 120 m/sec, and the others are accelerated forward and backwards symmetrically (or nearly so, for even number of satellites).

Fig. 2 presents the launch mass requirements of various propulsion options as a function of the number of microsatellites. The low Isp options perform poorly, because of the large ΔV required, and could at most be used for a constellation of 4 microsatellites. The MR508 arcjet is too powerful for this mission; a "scaled down" 290 W version, giving the same thrust as the SPT-70 (about 40 mN), would do significally better, but would still be marginal for 6 microsatellites. The SRM option is the best in this low Isp group, despite the inefficiency associated with using fully loaded STAR-13 engines (misalligned for the lower ΔV's); however, this option would lack the fine grain necessary for insertion into the near-Earth "trigger" orbit, and may present high-g problems.

Of the high Isp options, carrier-based XIPS, at 439W of power, would perform best, closely followed by the SPT-70, at 660 W. Because of the low power and high Isp, XIPS would take about 2.2 times longer to accelerate the microsats than SPT-70 (about 15 months). If not scaled to the same thrust as the SPT-70, these two systems perform indistinguishably. Both are perfectly capable of deploying 6 microsatellites, perhaps 7. It is interesting that a "scaled down" 100 W Hall thruster, with the SPT performance, would save about 10% additional launch mass, because the 70 Kg core carrier is not then acclerated.

The performance improvement of XIPS over SPT-70 is marginal. While over 15 SPT-70 thrusters have already been flown in space onboard Russian spacecraft [5], the XIPS has

not yet accumulated this amount of flight experience. Moreover, the SPT-70 thrusters are available at no cost from Phillips Laboratory as a result of an earlier mission's cancellation. The marginally better performance of XIPS was not sufficient to justify high cost and little flight experience and as a result, the readily available SPT-70 system was selected for the ETA mission.

ETA Baseline Mission Scenario

The ETA carrier spacecraft and six microsatellites will be launched in January 2000 aboard a DeltaLite launcher, which will place the stack onto an Earth escape trajectory as illustrated by Figure 3. The carrier spacecraft with microsatellite stack will then coast for 85 days before the SPT-70 propulsion system is fired for approximately 30 days. This thrusting period places the carrier spacecraft on a trajectory from where the two trigger satellites are deployed into their operational orbits. The trigger satellites orbit around the Earth at ranges between 0.020-0.036 AU, which facilitate high rate communications links for rapid GRB alert capability.

Having deployed the trigger satellites, the carrier spacecraft then thrusts in the retrograde (anti-velocity) direction for 69 days, after which the third microsatellite is deployed. Retrograde thrusting of the SPT-70 results in the carrier spacecraft getting closer to the Sun while it deploys microsatellites. The fourth, fifth and sixth microsatellites are deployed after inter-deployment thrusting periods of 58,50,43 days respectively. The shorter thrusting periods for later microsatellites are necessary to impart similar differential ΔV 's in order to result in a regularly spread constellation. After deployment of the sixth and final microsatellite about 11 months after launch, the carrier spacecraft is in a heliocentric orbit and can be utilized for post-deployment operations unrelated to the main ETA mission.

The differential ΔV 's imparted to the microsatellites cause their separation to increase angularly relative to each other. As a result, the constellation spreads out and begins to establish significant inter-spacecraft baselines. Figure 4 shows the heliocentric positions of the microsatellites at yearly intervals; orbit revolution is in the counter-clockwise direction. About two years after launch, the constellation will have developed into a configuration which provides the minimum baselines to commence nominal scientific operations.

Current plans call for science operations to continue for at least two years, resulting in a nominal mission duration of four years, including the two year launch, constellation deployment and development phase. The constellation can support GRB localization measurements to high accuracy beyond the nominal mission duration if extended science operations are contemplated.

SPT-70 System Hardware

The Stationary Plasma Thruster (SPT), also called a closed-drift Hall thruster, is an electromagnetic plasma device which achieves high Isp at low power with long electrode life. Since its first flight in 1971, over 50 units of the SPT-70 thruster have been flown onboard Russian spacecraft with no failures [5]. The West has been interested in using these proven thrusters and as a consequence, an American team of specialists conducted a series of tests on the SPT-100, both in Russia and in the US to validate its performance [6,7]. Extensive ground testing in the US has demonstrated the performance of the SPT-100, including testing to over 5,000 hours.

The US Air Force Phillips Laboratory acquired a few SPT-70 thrusters for testing on the Miniature Sensor Technology Integration (MSTI) spacecraft [8,9]. The thrusters could not be flown on MSTI-3 as planned and are available for the ETA mission through a mutual agreement [1]. The ETA SPT-70 system has been designed with hardware already developed or under development.

Two SPT-70 thrusters will be utilized onboard the ETA carrier spacecraft. They will be operated in a serial mode, effectively doubling the available thrusting time. Current design assumes a lifetime of 3,000 hours for each thruster. At the nominal operating point of 660 W, the SPT-70 produces 40 mN of thrust at an efficiency of 45% and Isp of 1510 sec. The thruster operates over a wide range of powers without substantial change in performance and this is important since the Sun range of the carrier spacecraft continues to change during the deployment phase.

Figure 5 presents the SPT-70 system block diagram [1]. Primary command/telemetry, power and propellant interfaces are shown. The two thrusters will provide redundancy and either of these can be operated through the Thruster Selection Unit (TSU), which is essentially a network of relays. The thrusters will be mounted on a gimbal to align the thrust vector along the carrier spacecraft's center-of-gravity, which will move as the microsatellites are deployed. The Power Processing Unit (PPU) is a derivative of the PPU developed by Space Systems/Loral for the SPT-100 [10]. It will operate at a constant discharge voltage of 300 V. The PPU is the central component of the system, providing power to the anode, magnet, cathode heater, ignitor and Xenon flow controller supplies and has an efficiency of 93%. It is also responsible for a number of control functions and general monitoring of thruster operation. The interface between the carrier and propulsion system was minimized by use of DCU and PDU. The Digital Control Unit (DCU) acts as the Command/Telemetry interface unit for the propulsion system and is the main controlling unit for the propulsion system. It accepts control commands from the spacecraft computer and command decoder unit and routes them to appropriate units such as the Propellant Management Assembly (PMA) and PPU. Telemetry, in the form of voltages, currents, temperatures, pressures and flow rates, from the various units is fed to the DCU which converts the data into a form appropriate for channeling to the spacecraft computer. The Power Distribution Unit (PDU) receives power from the main spacecraft bus at 42 V DC and distributes it to various components, notably the PPU, DCU, PMA and propellant tank heaters and valves.

The spherical propellant tank is sized for about 60 kg of Xe propellant which will be stored at 1000-1500 psi. The tank which is available off-the-shelf, is fabricated from stainless steel with graphite overwrap and is fitted with heaters for maintaining the propellant within the appropriate temperature range. The Propellant Management Assembly (PMA) consists of a series of valves, regulators, filters and pressure transducers to channel propellant from the tank to the Xenon Flow Controller (XFC) at the right flow conditions. The XFC is the flow control unit which serves a number of functions associated with thruster operation. Some of these include isolation between thruster firings, selection of the redundant cathode, flow throttling for regulation of discharge parameters and providing appropriate mass flow between anode and cathode (~90/10%) [11]. Taking advantage of the viscosity variations of Xenon with respect to temperature, a thermothrottle, which is essentially a heated capillary tube, controls flow rate by varying the temperature of the tube. Four additional valves were included in the system to allow for switching betwen the two SPT-70's.

At the nominal operating point of 660 W, the input power required at the PDU is about 790 W. Adding on a further 30 W for miscellaneous loads results in a net power requirement of about 820 W; the carrier spacecraft has a separate solar array to generate SPT-70 system

power. The propulsion system mass budget is given in Table 1. Total dry mass is estimated to be about 60 kg, with wet mass being about 120 kg. Figure 6 depicts the configuration of the propulsion system. This will be placed at the bottom of the carrier spacecraft, well away from the extended solar arrays.

Preliminary studies foresee no major problems associated with SPT-70/Carrier spacecraft interactions and spacecraft design has taken into account configuration issues related to plume contamination, sputtering and electromagnetic interference. A cold gas roll control system on the carrier spacecraft will be used to cancel out "swirl" torques generated during SPT operation. Nevertheless, further characterization tests of the SPT-70 are scheduled to be carried out at NASA Lewis Research Center. Phillips Laboratory will oversee SPT-70 system development and component testing. Integration with the carrier spacecraft will be done at the spacecraft contractor's facilities.

Conclusions

The propulsive requirements of the ETA mission require a high-Isp propulsion system, making it ideal for application of electric propulsion. Results of a trade study indicate that the SPT-70 system is well suited to meet ETA system requirements in terms of performance, cost, reliability, availability and flight heritage. The ETA baseline mission scenario has been developed taking into account operational characteristics of the SPT-70. The design and integration of the propulsion system is being led by Phillips Laboratory and incorporates hardware that is currently available or under development. SPT-70 characterization testing will be performed at NASA Lewis Center to better understand the operational characteristics of the system and thruster/spacecraft interactions. ETA is a unique opportunity to demonstrate the operation of electric propulsion technology in an interplanetary setting and the successful application of the SPT-70 thruster will pave the way for future implementation of EP systems for primary propulsion for all sorts of missions.

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Table 1: SPT-70 Propulsion System Mass Budget

	Quantity	Mass (kg)
Stationary Plasma Thruster	2	3.0
Power Processing Unit (PPU)	1	6.6
Digital Control Unit (DCU)	1	4.5
Thruster Selection Unit (TSU)	1	2.0
Propellant Management Assembly (PMA)	1	2.8
Power Distribution Unit (PDU)	1	7.5
Propellant Tank	1	11.3
Gimbal	1	6.0
Half- Xenon Flow Controller (XFC)	2	0.6
Latch Valves	4	0.4
Structure		5.0
Harness, Plumbing, Miscellaneous		5.0
Total Dry Mass		54.7
Margin		4.5
Xenon propellant load		60.0
Total Wet Mass		119.2

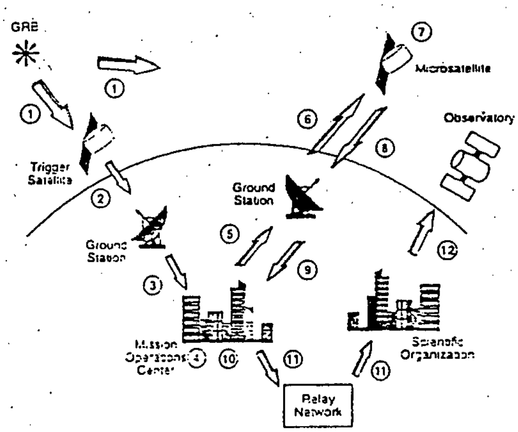

Figure 1: ETA System Concept

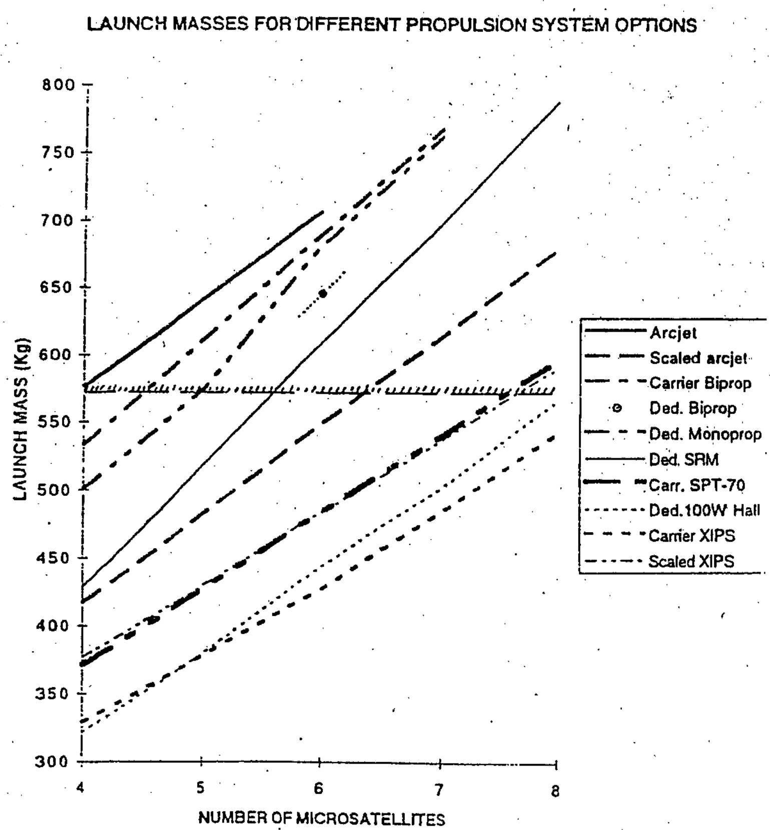

Figure 2: Launch Mass Requirements for Propulsion System Alternatives

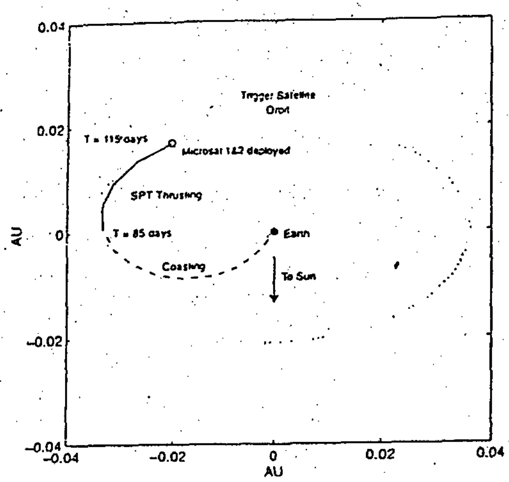

Figure 3: ETA Baseline Launch Trajectory

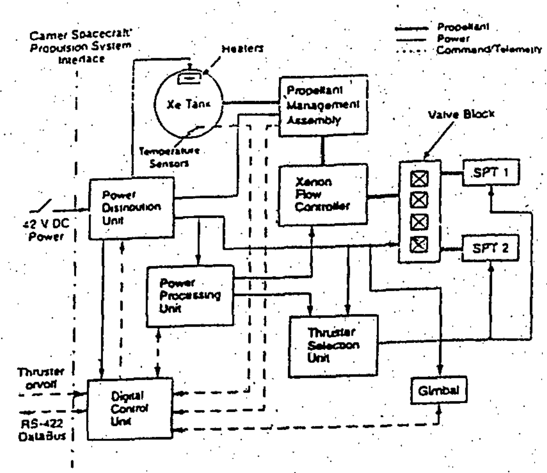

Figure 5: SPT-70 Propulsion System Block Diagram

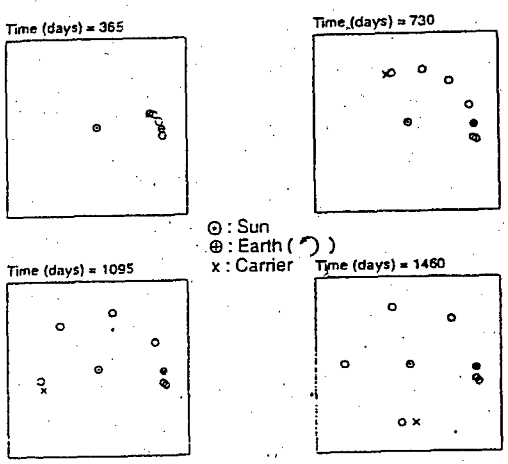

Figure 4: Constellation Development for ETA Baseline Scenario

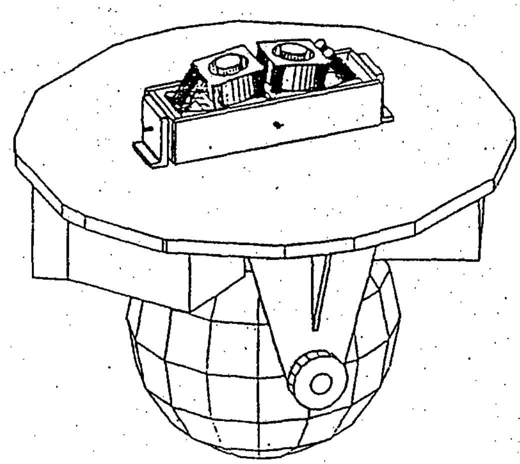

Figure 6: SPT-70 Propulsion System Configuration